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RESEARCH MEMORANDUM

FACTORS AFFECTING FLOW DISTORTIONS PRODUCED BY SUPERSONIC INLETS

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS
WASHINGTON
February 29, 1956

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SUMMARY

Typical effects of flow distortions on turbojet-engine performance are reviewed for the sake of completeness. Primarily, however, means are sought to reduce the distortions entering the compressor.

Flow distortions entering the compressor may be reduced by reducing the distortion entering the inlet diffuser and by improving the amount of mixing that generally occurs in the subsonic diffuser. Sources of distortion in the supersonic inlet are enumerated and steps are suggested that might be taken to reduce their effects. Also, parameters affecting the mixing of these distorted flows in the subsonic diffuser are discussed.

INTRODUCTION

The performance of many current subsonic and transonic airplanes has been reduced to some extent by the presence of nonuniform flow at the compressor of the turbojet engine. These flow distortions are characterized by variations in the velocity or total pressure of the air entering the compressor and are usually expressed either as the maximum variation in velocity ΔV or total pressure ΔP divided by a reference velocity or pressure.

Typical effects of distortion on turbojet engine performance are discussed for completeness. It is evident that these flow distortions must be held to a minimum, and applicable factors are discussed. Sources of distortion in the supersonic inlet are enumerated and steps are suggested which might be taken to reduce their effects. In addition parameters affecting the damping out of these flow distortions in the subsonic diffuser are discussed.

SYMBOLS

The following symbols are used in this report:

D	diameter of constant-area straight section
D_h	hydraulic diameter of inlet throat
H_i	inlet-throat height
h/δ	parameter defining amount of external boundary layer removed 4011
L	length of subsonic diffuser
l	length of constant-area straight section
M	Mach number
m	mass flow
P	total pressure
ΔP	difference between maximum and minimum values of total pressure as measured by rake
Δp	static-pressure increment
q	dynamic pressure
V^*	stagnation speed of sound
ΔV	difference between maximum and minimum velocities
δ	boundary-layer thickness
θ_c	cone half-angle

Subscripts:

av	average
b	annulus
0	free stream

EFFECT OF DISTORTIONS ON ENGINE PERFORMANCE

The penalty of flow distortion on engine performance is dependent upon the particular engine under consideration and upon the type of flow distortion (that is, whether the flow varies radially between the compressor hub and blade tip, or varies circumferentially around the compressor annulus, or, as is the usual case, whether the flow has components of both radial and circumferential distortion). Also the magnitude and extent of the flow distortion affects engine performance. Therefore, in the following discussion, some of the possible effects of distortion on engine performance will be discussed. It should be understood that any given engine-duct combination may not suffer from all of the observed effects; on the other hand no engine-duct combination has been found immune from all of these effects.

A typical effect of circumferential distortion is shown in figure 1. When circumferential distortion enters the compressor, the distortion, although reduced in magnitude, persists through the last compressor stage. Temperature gradients then exist at the turbine as the result of circumferential variation in the fuel-air ratio caused by the distorted air flow. Although the turbine rotor feels only the average temperature, the turbine stator at some point is subjected to higher than the average temperatures. In order to prevent failure of the stator due to overheating, the average turbine temperature must be reduced (refs. 1 to 3).

The amount of necessary turbine-temperature reduction for several engines is shown in figure 1 as a function of the entering total-pressure distortion. Reducing the turbine temperature as required results in the indicated net thrust losses. These reductions in engine thrust can amount to as much as 1 percent for each 2 percent of total-pressure distortion.

Another effect of flow distortion is that of reducing the stall margin of the compressor. This effect is illustrated in figure 2 for two typical engines. Compressor pressure ratio is plotted as a function of the corrected engine speed. The presence of radial distortion lowered the compressor surge limit line in comparison with the steady-state operating line and caused the rotating stall region to move to higher corrected engine speeds (fig. 2(a)). (See, for example, refs. 1 and 3.) Lowering of the surge limit line reduces the acceleration margin of the compressor. Movement of the rotating stall region to higher corrected engine speeds indicates that the maximum flight speed at altitude may be reduced by the occurrence of rotating stall.

The presence of circumferential distortion for another typical engine again lowered the compressor surge limit line (fig. 2(b)). The acceleration margin of the compressor is again reduced. For distortions of the order of 32 percent, the maximum possible corrected engine speed is about 108 percent. Hence, the cruising speed of the airplane can be affected by the occurrence of large circumferential distortions.

Another typical effect of flow distortion is the reduction of maximum altitude due to compressor surge (for example, ref. 1). An example of this effect is presented in figure 3. Altitude is plotted as a function of the compressor speed. With uniform flow at the compressor, a maximum altitude of about 62,000 feet was achieved. This limit is primarily a result of Reynolds number effect. At the higher compressor speeds, the altitude was limited by the maximum turbine-outlet temperature.

When circumferential distortion was introduced into the compressor, the maximum altitude was reduced to as low as 47,000 feet. Although the distortion level was about 20 percent for both cases presented, quite different effects on the altitude limits were observed. The greatest reduction in altitude occurred with the extended circumferential distortion rather than the localized distortion. In order to get the compressure out of surge for these cases, a drop in altitude to 35,000 feet was necessary.

Finally, distortion increases the stresses and vibrations of the compressor and may reduce engine cyclic efficiencies (ref. 1 to 3).

In order to reduce these performance penalties to a minimum, it is essential that the air inlet produce as low a distortion at the compressor face as possible.

DISTORTION PRODUCED IN SUPERSONIC INLETS

Intuitively, the distortion existing at the exit of the subsonic diffuser might be said to depend upon the amount of distortion entering the inlet throat, upon the existence of additional sources of distortion within the inlet, and upon the amount of mixing, or damping, of these distortions in the subsonic diffuser.

Mixing in Subsonic Diffuser

The amount of mixing that takes place in the subsonic diffuser is known to be a function of the length of the diffuser. For example, the total-pressure distortions at the diffuser exit of a variety of side-inlet types and for Mach numbers ranging from 1.5 to 3.0 are presented in figure 4(a) as a function of the ratio of diffuser length L to throat hydraulic diameter D_h. The data points represent the distortion for either critical or engine-inlet matching conditions. These distortions were obtained from the data of references 4 to 10 and from various unpublished data.

Although there is considerable scatter of the data, there is a definite trend of lower distortions with increased diffuser length. This scatter is due, in part, to the difference of distortion existing at the inlet throat, a factor which is considered in the section Distortion at Inlet Throat. Also, part of the scatter is due to the difference in average Mach number of the ducts. For example, when mixing takes place in a straight duct of constant area at a constant Mach number (ref. 11), as shown in figure 4(b), a smooth curve, similar to that sketched in
4011 through the data points, results, although the parameters L/D_h and l/D are not strictly comparable.

The effect of average Mach number on mixing is considered in figure 5. In figure 5(a), the theoretical variation of the velocity distortion is plotted as a function of the average flow Mach number for three values of total-pressure distortion. In this example, velocity distortion is arbitrarily defined as the difference between the maximum and minimum velocity ΔV divided by a constant velocity, in this case the stagnation speed of sound V^*. At the lower Mach numbers the velocity distortion increases for a given level of total-pressure distortion. Thus, increased mixing may be expected to occur in a duct at the lower flow Mach numbers.

Also, for a given average Mach number, the velocity distortion decreases as the total-pressure distortion decreases. Thus, the decrease of distortion by mixing is proportional to the distortion. For example, the mixing that occurs in a straight duct of constant area at a given flow Mach number would tend to decrease the total-pressure distortion asymptotically toward some minimum value. There are some indications that this minimum value is probably that determined by the flow profile for fully developed pipe flow and would increase as the Mach number increases.

To illustrate some of these effects, total-pressure distortions measured in a constant-area duct at flow Mach numbers of 0.20 and 0.37 are presented in figure 5(b). At the lower Mach number, distortion decreases fairly rapidly through mixing. However, the distortion decreases less rapidly at the higher Mach number because of both the decreased rate of mixing and the decreased residence time in any given length of duct. The lowest value of distortion that could be expected with longer mixing lengths would be about 2 percent at the flow Mach number 0.2 and about 7.5 percent at Mach number 0.37 based on a fully developed turbulent profile. All distortions hereinafter are presented in terms of total-pressure rather than velocity.

The amount of mixing that occurs in the inlet subsonic diffuser will depend upon the average Mach number of the duct, inasmuch as the flow is diffused from a relatively high Mach number at the inlet throat to a lower Mach number at the diffuser exit. An example of the importance of low average Mach numbers in the diffuser duct is presented in figure 6. The distortion at the diffuser exit for critical inlet operation is plotted against free-stream Mach number for two nose-inlet models which were identical with the exception that the cowl and center-body surfaces were altered to give different rates of initial area expansion at the inlet lip. These data, although not published, were obtained from the investigations of references 12 and 13.

Although the distortion at the inlet throat was identical for both models, the model with 12 percent initial area expansion per inlet-throat hydraulic diameter had lower distortion at the diffuser exit than the model with the smaller area expansion. The two models discharged at the same Mach number and, of course, had the same Mach number at the inlet lip. The variation of Mach number through the ducts, however, was different because of the change in initial area expansion. The model with the larger area expansion diffused more rapidly, giving a lower average duct Mach number. Thus, the lower distortion is believed to be the result of more efficient mixing in the subsonic diffuser. 4011

There is evidence from these same data that constant-area throat inlets, which have proven desirable in some cases from stability considerations, and internal contraction inlets, which promote low cowl drags, will have higher distortions as the result of reduced mixing. From figure 6, mixing is promoted by rapidly expanding the diffuser duct. Carrying this concept a little further, lower distortions apparently could be produced by overexpanding the diffuser duct area to provide low Mach numbers to increase the mixing. The flow could then be rapidly accelerated at the diffuser exit to the desired Mach number. This concept is feasible inasmuch as the use of rapid acceleration will, of itself, reduce distortion. Flow acceleration has been known for some time to achieve more nearly uniform flow at the throats of subsonic wind tunnels. An example of rapid flow acceleration at the diffuser exit is shown in figure 7. Total-pressure distortions were measured in a constant-area straight duct with and without the benefit of rapid acceleration. Acceleration from Mach number 0.37 to 0.50 in the annulus was provided by a hub simulating the accessory housing of the turbojet engine. Insertion of the hub decreased the total-pressure distortion from about 16 to about 12 percent at the end of the straight section with essentially no loss in total-pressure recovery.

Forced Mixing Devices

The use of freely rotating blade rows to reduce flow distortion at low flow velocities is reported in reference 14. The use of such blade rows has since been investigated theoretically and the results have appeared promising in that distortion reduction can be achieved with little or no total-pressure loss. Such a blade is free wheeling at a

speed determined by the blade angle and average axial Mach number and reduces distortion by acting as a turbine in the higher than average velocity flow and as a compressor in the low velocity flow. The blade row then transfers energy to the low velocity region with no net work except for that required to overcome bearing friction. A model of the freely rotating blade-row apparatus has been built and tested, and the results are presented in figure 8. These data were obtained from unpublished tests. A blade row and a row of straightener vanes were mounted on a hub in a straight duct. Distortion was introduced by
4011 throttling the duct flow across screens placed in a portion of the forward duct. The distortion in the annulus at B was then measured and plotted as a function of the distortion at A ahead of the hub. The variation of the Mach number in the annulus due to the throttling is also plotted on the abscissa.

The distortion in the annulus is lower than that ahead of the hub for any point below the slanted line through the origin of coordinates. When circumferential distortion was introduced, some distortion reduction was achieved without the benefit of the blade row. This reduction was the result of the flow acceleration provided by the hub and was not affected by the row of flow straighteners. When the blade row was added, the distortion in the annulus was further reduced. When radial rather than circumferential distortion was introduced, about the same distortion reduction was achieved across the blade row. At an annulus Mach number of 0.5, the total-pressure distortion was reduced from 25.5 to 15 percent across the blade row. For this reduction in distortion the total-pressure loss was about 2.5 percent.

Screens have also been used as forced mixing devices (for example, refs. 15 to 17). An example of their use is shown in figure 9. The total-pressure distortion was measured behind screens of varying solidity, that is, the blockage area of the screen expressed as a percentage of the duct cross-sectional area. Two typical applications are considered; the curve for the lower Mach number 0.20 is representative of a ram-jet application, whereas that for the higher Mach number 0.50 is more typical of the diffuser discharge Mach number of a present-day turbojet engine; the Mach number ahead of the screens was about 0.20 and 0.37, respectively. For both cases, increasing the screen blockage reduced the distortion. However, there is a limit to the amount of screen blockage that may be added with no choking at the screens. With choking, of course, inlet mass flow would be reduced.

The total-pressure loss across the screens is plotted again as a function of the screen solidity (fig. 9(b)). At a solidity of 30 percent, the total-pressure loss was 2 percent at the low average Mach number and about 8 percent at the higher Mach number. In terms of the static-pressure increment divided by the upstream dynamic pressure $\Delta p/q$, these losses were about 0.75 and 1.50, respectively, for the 30 percent blockage screen.

In order to propel future aircraft to higher supersonic speeds, higher weight-flow turbojet engines will be required. For such engines the diffuser discharge Mach numbers will be of the order of 0.6 or even higher over portions of the flight range. Hence, mixing in the subsonic diffuser will probably be reduced. Forced mixing devices will be used with caution inasmuch as higher total-pressure losses have been indicated at higher duct Mach numbers. For such configurations, the distortion entering the inlet throat must be kept to a minimum.

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Distortion at Inlet Throat

Distortion at the inlet throat is caused primarily by nonuniform compression. Some of the origins of these distortions are examined and the resulting distortions at the diffuser exit are presented in figures 10 to 12.

The total-pressure variation across the inlet throat was determined for the full-scale J34 nose inlet at Mach number 1.8 (fig. 10). These data were obtained from unpublished data related to reference 18. Total-pressure profiles at the inlet throat are presented as a function of inlet mass-flow ratio. For the indicated configuration, a vortex sheet originating at the intersection of the oblique and normal shocks enters the inlet throat for values of mass flow less than the critical value. Theoretically, the air entering the inlet next to the cowl will be at a pressure recovery of about 81 percent, whereas the air below the vortex sheet is at a pressure recovery of about 96 percent. As indicated, the measured total pressures across the vortex sheet agreed quite well with the theoretical values. As the mass flow was reduced, the vortex sheet progressed farther toward the cone surface.

The distortion at the inlet throat was determined from these profiles between the indicated limits to eliminate the effects on the boundary layer and the results are presented in figure 11. As the mass flow was reduced, inlet distortion increased because of farther entry of the vortex sheet. Presented for comparison is the distortion measured at the diffuser exit. As the inlet mass flow was reduced, the exit distortion increased corresponding to the increase of distortion at the inlet throat. However, as the mass flow was reduced, the average Mach number of the flow through the diffuser decreased. Hence, at the lower mass-flow ratios the distortion at the diffuser exit decreased because of increased mixing, although the inlet distortion remained high.

The entrance of the vortex sheet, then, increases the distortion level of the diffuser. By delaying entry of the vortex sheet, lower distortions can be achieved. This may be done, for example, by positioning the oblique shock ahead of the inlet lip so that the vortex sheet will pass around the inlet for subcritical inlet operation rather than into the inlet. With variable-geometry inlets having internal contraction sufficient to choke the inlet, the entry of a vortex sheet is unavoidable and, as a result, the distortion entering the inlet is apt to be large. Moreover, mixing in the subsonic diffuser of inlets with internal contraction will probably be reduced because of the higher inlet-throat Mach numbers.

Another example of nonuniform compression is that occurring at angle of attack (fig. 12). The distortion was measured at the inlet throat, at two intermediate positions, and at the diffuser exit for the full-scale J34 conical-nose inlet at Mach number 2.0 for critical inlet operation.

Theoretically, at zero angle of attack, 3 to 4 percent distortion would be expected at the inlet throat because of the nature of the conical flow field. Seven-percent distortion, however, was measured. The higher distortion is due to boundary-layer effects which were not considered. Similarly, at an angle of attack of 10°, nonuniform compression would be expected to yield inlet-throat distortions of the order of 14 to 15 percent around the inlet-throat circumference. The actual distortion measured, even at an angle of attack of 3°, was over twice this theoretical shock value, again because of shock-boundary-layer interaction. Along the upper surface of the cone, the Mach number behind the oblique shock was quite high because of the reduced compression. The terminal shock was then strong enough to separate the compression surface boundary layer and to increase the distortion.

As the flow traversed the diffuser, distortion decreased because of mixing. However, the mixing was not sufficient to overcome the large initial distortion, and the distortion increased at the diffuser exit as angle of attack was increased.

At zero angle of attack, the distortion measured at the second measuring station was larger than at the inlet throat. This increase was due to separation of the boundary layer from the centerbody surface between the two measuring stations. Thus, sources of distortion exist within the inlet. Separation, rapid duct turns, struts, and so forth are all possible sources of distortion within the inlet.

In order to reduce the characteristic increase of distortion in conical-nose inlets at angle of attack, such inlets must be shielded from angle-of-attack effects. For example, the inlets could be located beneath and behind the wings and close to the fuselage. Preliminary test results indicate also that alinement of the spike centerbody with the free-stream direction reduces the nonuniform compression and hence reduces distortion.

Other inlet types may be required to reduce distortion at angle of attack. For example, horizontal ramp inlets are less sensitive to angle of attack, inasmuch as changes in angle of attack merely change the effective angle of compression.

The occurrence of shock-boundary-layer interaction was seen to have an important effect on distortion in figure 12. This factor is examined more closely in figure 13. The flow in the throat of a conical-nose inlet for critical inlet operation at a Mach number of 1.8 was determined for a range of values of the conical compression angle θ_c (ref. 18). By varying the compression angle, the Mach number ahead of the terminal shock was changed.

The thickness of the compression surface boundary layer δ expressed as a percentage of the inlet height H_i is shown in figure 13(a). Between cone angles of 30° and 25°, the boundary-layer thickness changed only from 3 to 4 percent of the inlet-duct height. For the 20° cone half-angle, however, the boundary-layer thickness doubled because of separation at the terminal shock. The occurrence of this separation had been predicted theoretically on the basis of the static-pressure rise across the terminal shock.

The distortion at the inlet throat is plotted as a function of the cone half-angle (fig. 13(b)). Although the distortion did not change for the higher cone angles, the distortion increased considerably for the separated flow case at a cone half-angle of 20°.

The distortion measured at the diffuser exit is again plotted for comparison. As the cone half-angle was decreased from 30° to 25°, the average-flow Mach number of the duct decreased as a result of the reduced throat Mach numbers, and the exit distortion decreased although the distortion at the throat remained about the same. At θ_c of 20°, the increased mixing could not overcome the high initial distortion, and the distortion at the exit increased somewhat.

Suitable control of the compression-surface boundary layer would be expected to reduce these separation effects. For example, in figure 14 a typical ramp side inlet had provisions for both external and internal boundary-layer removal. With no external removal, the distortion at the diffuser exit varied between 40 and about 63 percent for critical inlet operation. By moving the inlet out of the boundary layer, the distortion was progressively decreased until, for complete external removal, the distortion was in the 15 to 19 percent level. The lowest distortion was measured when internal boundary-layer removal was used in conjunction with external removal. For these cases, the distortion was reduced to a more acceptable level of 7 to 10 percent. Thus, suitable removal of the boundary layer with side inlets is essential not only from the standpoint of increasing the pressure recovery but also of decreasing the distortion.

CONCLUDING REMARKS

Distortion at the compressor can be reduced by reducing the distortion entering the inlet throat, by eliminating internal sources of distortion, and by improving the mixing in the subsonic diffuser.

The distortion that exists at the inlet throat is primarily a result of nonuniform compression and may result from the entrance of the vortex sheet, operation at angle of attack, shock-boundary-layer interaction, or combinations of these effects. A few steps can obviously be taken to reduce these sources of distortions. For example, entrance of the vortex sheet can be delayed by positioning oblique shocks ahead of the inlet lip. Shock-boundary-layer interaction can be controlled to some extent by the use of boundary-layer control such as compression surface bleed.

With conical-nose inlets, large distortions at angle of attack are indicated. Low distortions can be achieved only by shielding such inlets from angle of attack. Other than conical compression inlets may be required to reduce angle-of-attack effects. For example, horizontal ramp inlets are less sensitive to angle of attack.

The mixing of distorted flows in the subsonic diffuser is primarily a function of the length and the average Mach number of the duct. Additional mixing can be obtained with forced mixing devices but generally at the expense of pressure recovery and weight. This brief review of the problem would indicate that severe distortion problems can be expected in the future with high compression inlets and high weight-flow engines required for flight at the higher supersonic speeds. With such configurations the average duct Mach number will be high and little mixing will occur in the subsonic diffuser. The flow distortion entering the inlets must thus be kept to a minimum, and sources of distortion within the inlet must be eliminated. No simple solution to the distortion problem is currently evident. Satisfactory duct-engine combinations will require careful attention to detail and perhaps compromises in both the airframe and the engine designs.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, November 1, 1955

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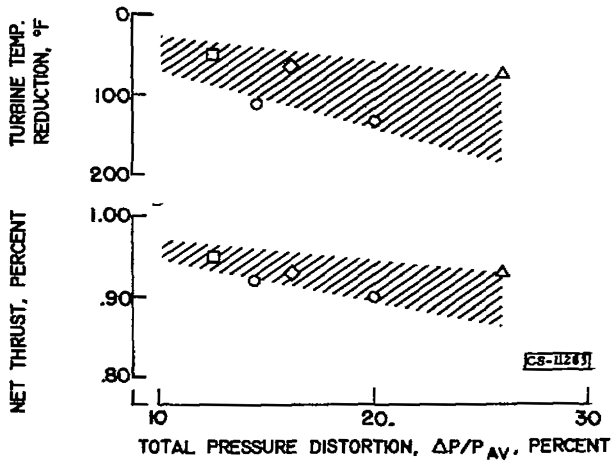
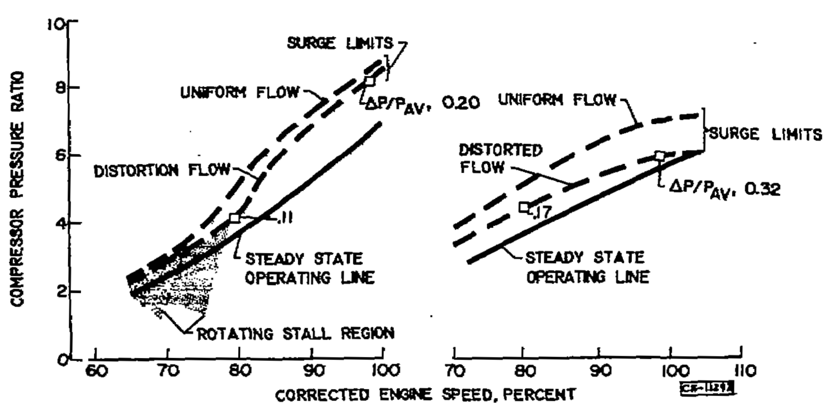


Figure 1. - Performance derating with circumferential distortion.



(a) Radial distortion. (b) Circumferential distortion.

Figure 2. - Effect of distortion on surge limits.

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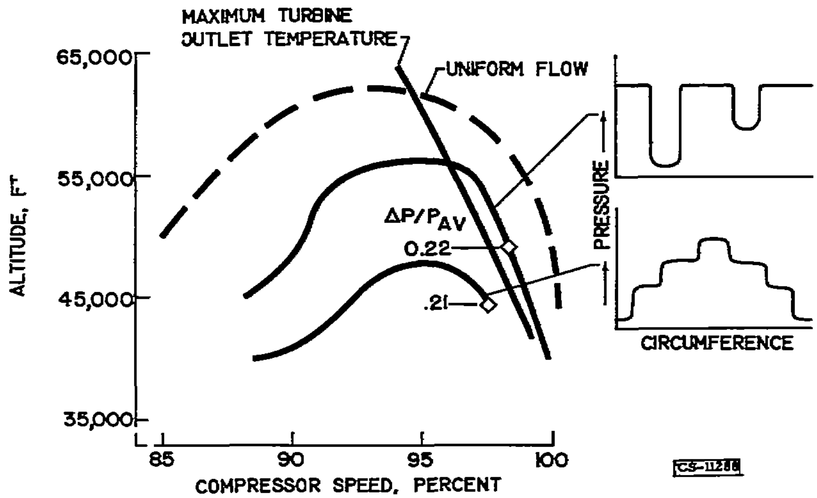
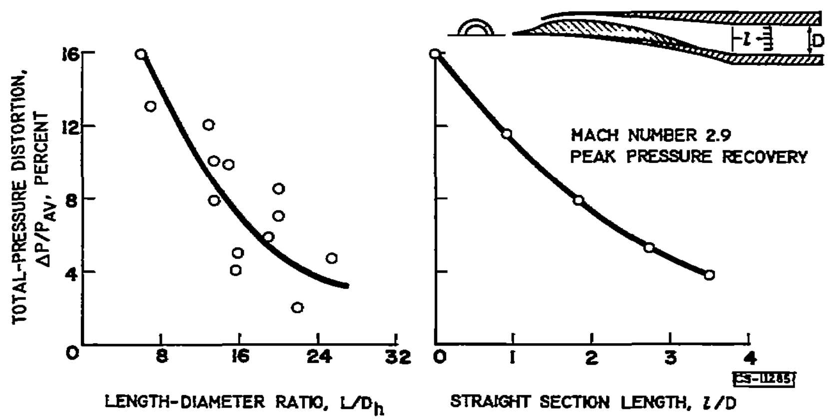


Figure 3. - Effect of distortion on surge at altitude.



(a) Data of references 4 to 10. (b) Data of reference 11.

Figure 4. - Effect of mixing length on distortion.

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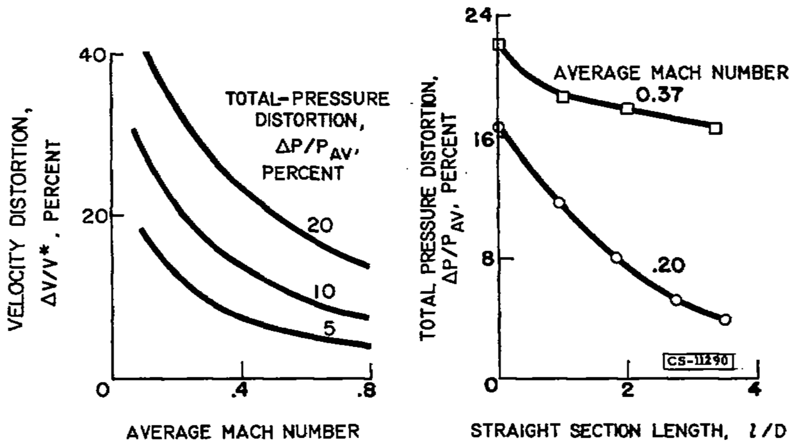

(a) Theoretical. (b) Experimental.

Figure 5. - Effect of average flow Mach number on mixing.

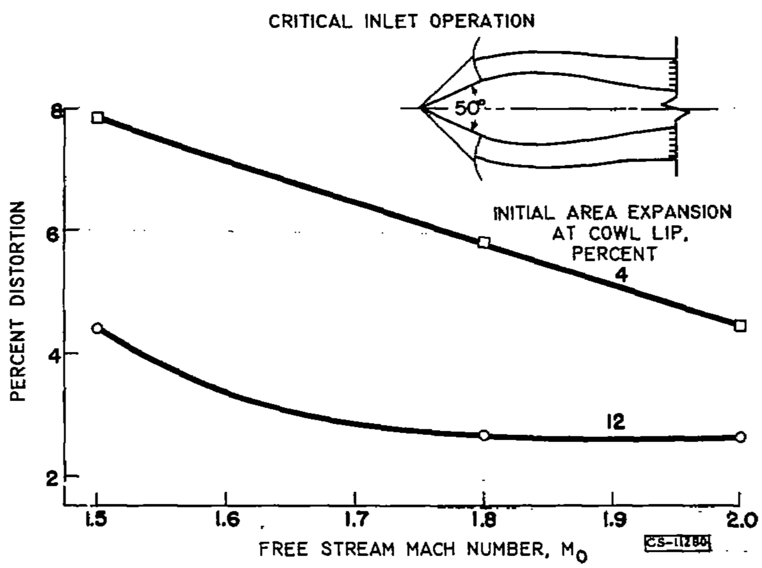

Figure 6. - Effect of cowl area distribution on distortion.

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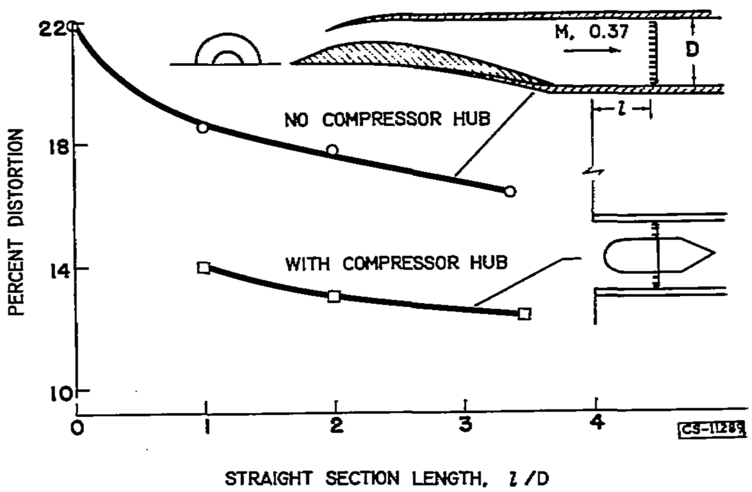


Figure 7. - Effect of flow acceleration on distortion. Mach number, 1.9.

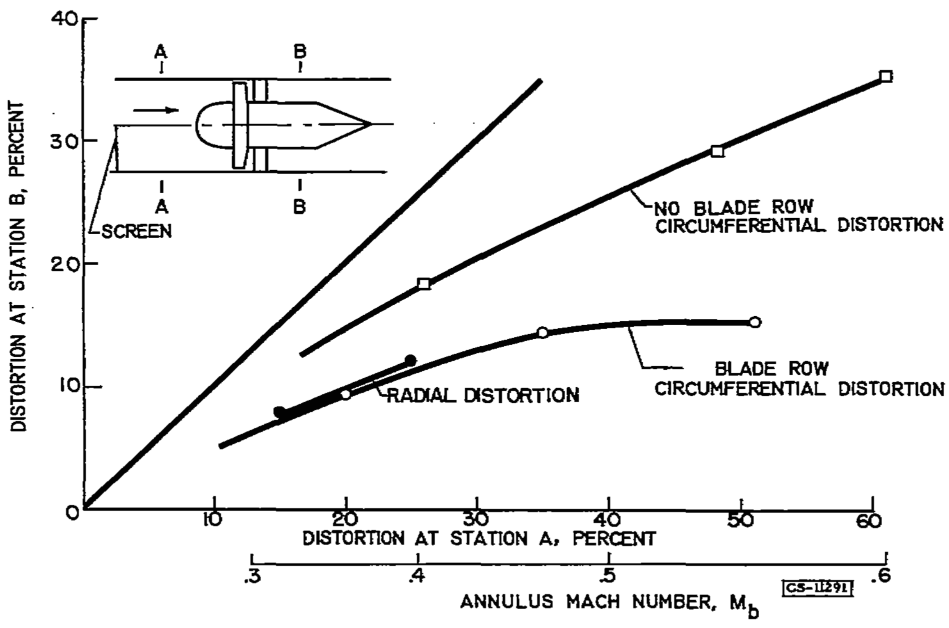
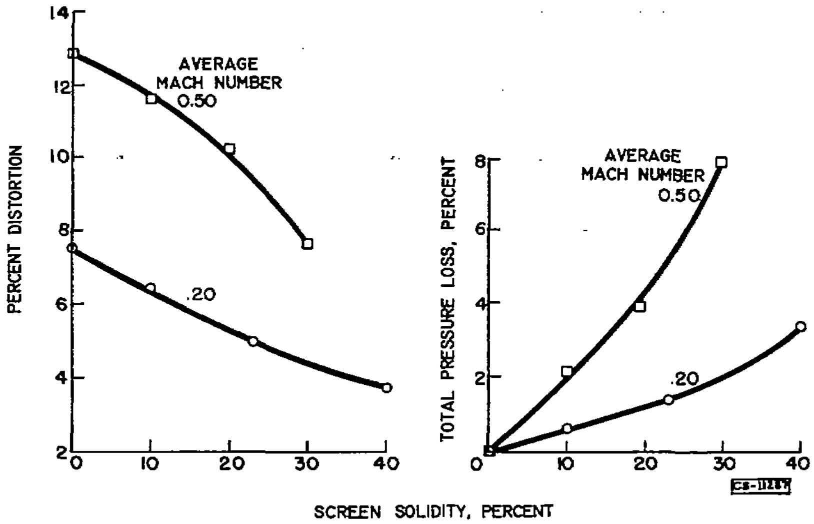


Figure 8. - Effect of freely rotating blade rows on distortion.



(a) Distortion. (b) Total-pressure loss.

Figure 9. - Effect of screens on distortion.

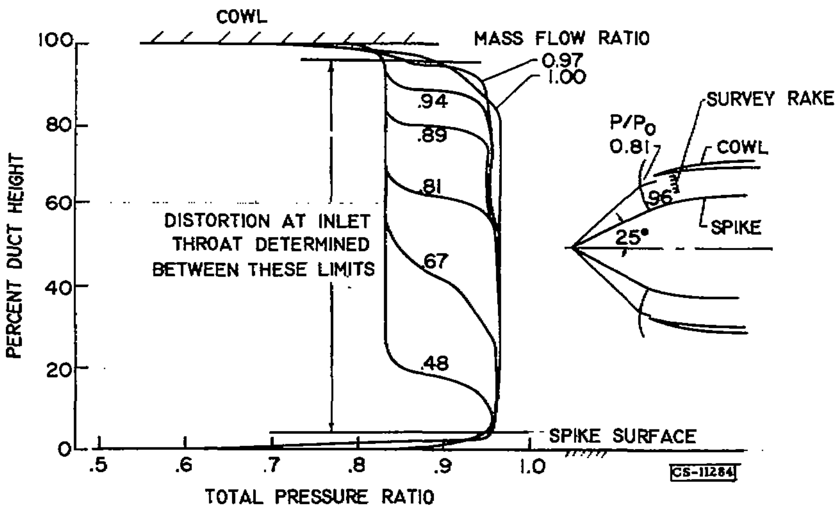


Figure 10. - Effect of vortex sheet on distortion. Mach number, 1.8.

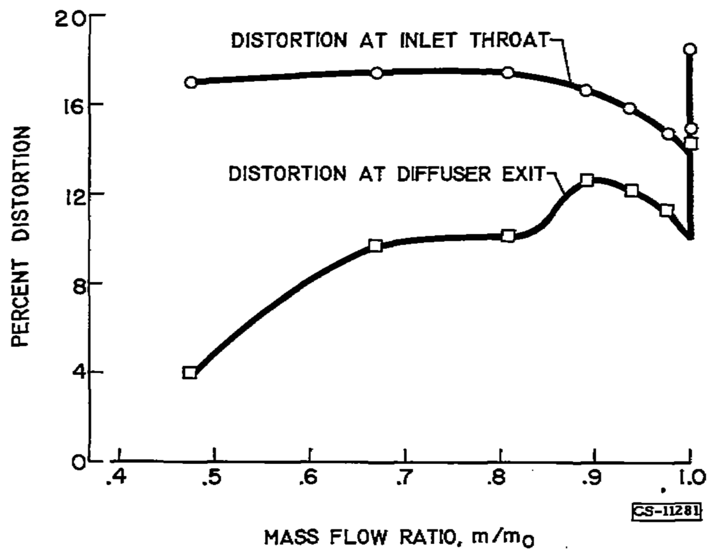


Figure 11. - Effect of vortex sheet on distortion. Mach number, 1.8.

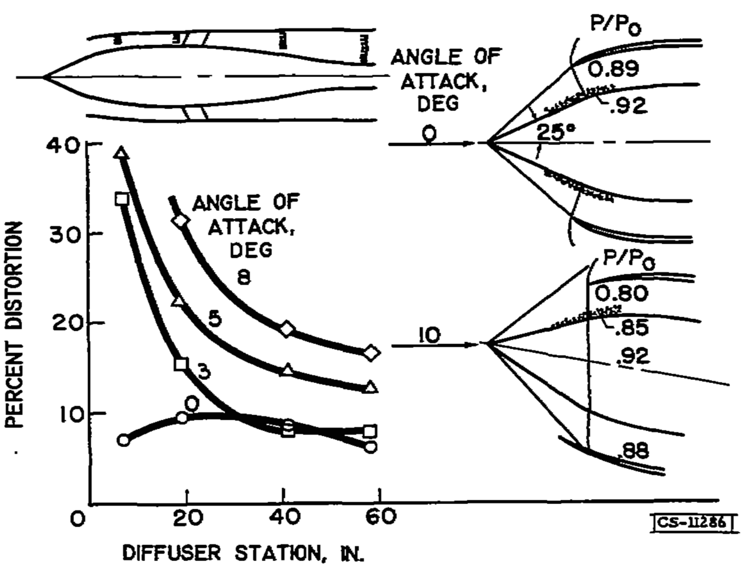
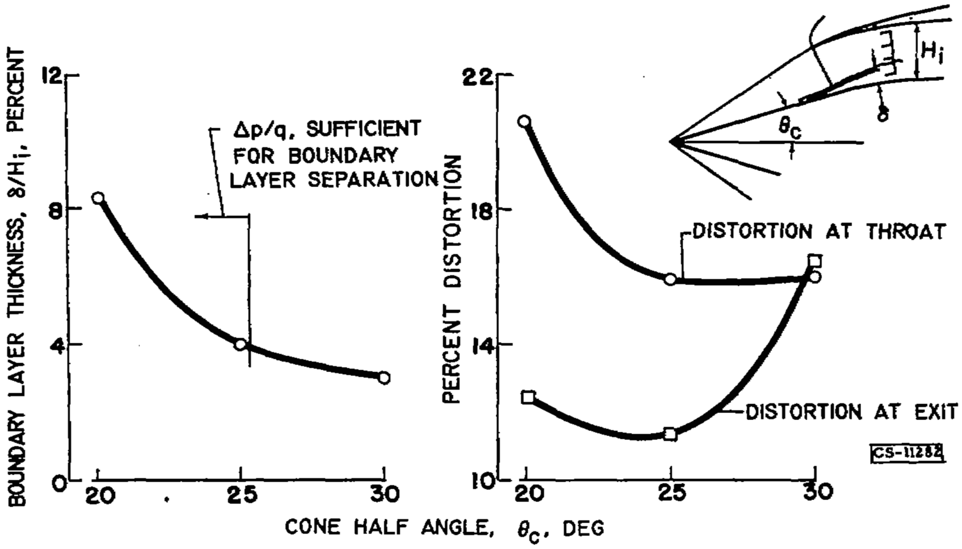


Figure 12. - Effect of angle of attack on distortion. Mach number, 2.0.



(a) Boundary-layer thickness. (b) Distortion.

Figure 13. - Effect of shock boundary-layer interaction. Mach number, 1.8.

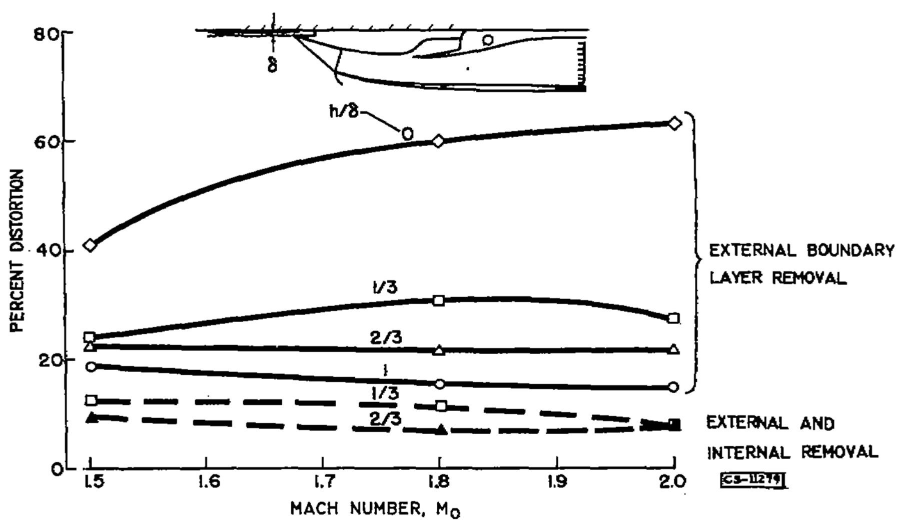


Figure 14. - Effect of boundary layer removal on distortion.

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